

Introduction to Solid Rocket Propulsion

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SUMMARY

The objectives of this introduction are to present the fundamentals of solid rocket motor (SRM), starting from the elementary analysis of rocket operation and then justifying the need of sophisticated computation of the internal flow. After a brief reminder of solid rocket history, a description of its main components is proposed. The elementary parameters controlling the operation are introduced and the basic formula predicting the steady-state operation pressure is established. The main issues faced by a SRM require an accurate description of internal aerodynamics, either to predict the pressure/thrust programs and the normal transient phases like ignition, or to study the motor stability. A short overview of the evolution of the codes devoted to SRM internal aerodynamics during the last thirty years is given in order to introduce the more specialized presentations; a discussion of the main limitations concerning these codes is also proposed. The prospects offered by SRM internal aerodynamics codes are finally described.

1.0 GENERALITIES

1.1 History

The solid rocket motor belongs to the family of the rocket engine (thrust achieved by mass ejection) and its history can be considered both ancient and recent. It is possible to consider that the black powder is the precursor of modern solid propellants: composed of natural ingredients (sulfur, charcoal and saltpetre), the black powder has been used from the 13th century in Asia to propelled darts, certainly the first unguided stand-off weapons. A lot of work has been performed since this time to improve the solid propellant and to master its combustion but the main military application has been gun propellants up to the WW2. The WW2 has seen the first aeronautical applications (BACHEM Natter, JATO, RATO). The main developments for military (missiles) and space activities (launchers) started in 1945.

Regarding the space activities, the first flights were carried out by liquid propellant rockets, following the world's first successfully flown rocket on March 15, 1926 (R. Goddard, USA). The first satellites have been put into orbit by a liquid propellant launcher (R7 Semiorka, October 1957); the first successful US launch (Jupiter C, January 1958) used solid propellant rockets for the upper stages. The small US Scout has been the first all solid propellant launcher. Most of the first intercontinental missiles or intermediate range missiles used also liquid propellant engines, for their first generations.

The current situation is the following:

- Most of the modern strategic and tactical missiles use solid propellant propulsion. The only competitor for solid propulsion is ramjet propulsion for tactical missiles.
- Space launchers are in the western countries and in Japan based on an assembly of liquid and solid propelled stages; they remain all liquid propellant in Russia, Ukraine and China. This difference of design is clearly connected to economical considerations: development and

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recurrent costs of a large solid propellant booster are lower than those of a large liquid propellant booster in the western countries, agreed that performance is better for liquid propellant propulsion. The orientation towards RLVs (Reusable Launch Vehicles) will favor of course liquid propulsion in the future.

1.2 The Basic Solid Rocket Motor

A solid propellant rocket is formed by four main components (fig. 1):

- A case containing the solid propellant and withstanding internal pressure when the rocket is operating.
- The solid propellant charge (or grain), which is usually bonded to the inner wall of the case, and occupies before ignition the greater part of its volume. When burning, the solid propellant is transformed into hot combustion products. The volume occupied by the combustion products is called combustion chamber.
- The nozzle channels the discharge of the combustion products and because of its shape accelerates them to supersonic velocity.
- The igniter, which can be a pyrotechnic device or a small rocket, starts the rocket operating when an electrical signal is received.

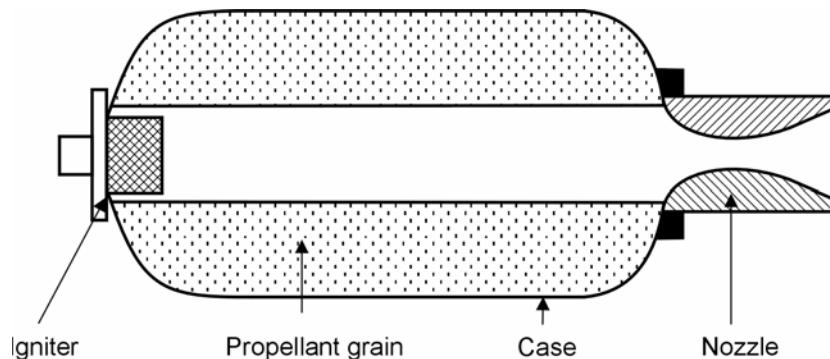


Figure 1: Basic Solid Rocket Motor.

One can consider that the solid propellant after manufacturing is in a metastable state. It can remain inert when stored (in appropriate conditions) or it can support after ignition its continuous transformation into hot combustion products (self-combustion). The velocity of the transformation front is called *burning rate* (fig. 2).

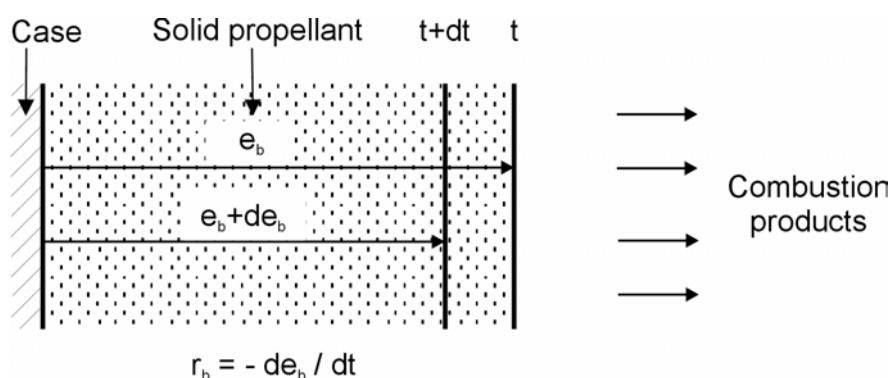


Figure 2: Solid Propellant Rocket Motor.

The solid rocket is therefore inherently simple and therefore can possess high intrinsic reliability. After ignition, a solid rocket motor normally operates in accordance with a preset thrust program until all the propellant is consumed. All the efforts should be directed to the accurate prediction of the thrust (and pressure) programs to get the benefit of solid rocket motor concept.

1.3 Definitions

$$\text{Efficiency of a rocket if defined by specific impulse: } I_s = \frac{F}{\dot{m}g_o}$$

where: I_s is the specific impulse (in s),

\dot{m} is the mass flow rate ejected by the rocket (in kg.s^{-1}),

g_o is the reference gravity ($9,806 \text{ m.s}^{-2}$).

This definition has been adopted because the second is a universal unit of measure. Specific impulse is used for all types of rocket (chemical, electric) and for ramjet/scramjet; specific impulse can be considered as the inverse of specific consumption. The higher is the specific impulse, the better is the rocket.

It should also be observed that specific impulse depends on the operation conditions of the solid rocket motor: pressure in the combustion chamber, rate of expansion in the nozzle, ambient pressure. The theoretical specific impulse, for a given propellant and a fixed pressure, can be predicted by thermochemical computation. All countries in the world are using similar codes and, in the western countries, the same thermodynamic tables (JANAF). To avoid all the problems related to unit systems, standard conditions are adopted and comparisons can be performed:

- Pressure: 70 atm (close to 1000 psia).
- Expansion: 70 to 1 atm.
- Ambiant pressure: 1 atm (nozzle adaptation).

Under these reference conditions, the theoretical specific impulse of a conventional composite propellant (ammonium perchlorate, polybutadiene, aluminum) is around 265 s. In practice, the actual specific impulse is lower, due to specific impulse losses related to phenomena which cannot be taken into account in the thermochemical computation (multidimensional flow, two-phase effects, wall phenomena...). Specific impulse in vacuum is higher than I_s at atmospheric pressure.

Another characteristic parameter, essentially defined for a chemical rocket equipped with a choked convergent-divergent (de Laval) nozzle, is the *characteristic velocity*:

$$C^* = \frac{p_o A_c}{\dot{m}} (\text{m.s}^{-1})$$

- where p_o is the pressure in the combustion chamber (Pa),
- A_c is the nozzle throat area (m^2).

A simplified description of the flow in the nozzle (one-dimensional flow, ideal gas) gives a clear interpretation of C^* :

$$C^* = \frac{\sqrt{\frac{R}{M}} T_o}{\Gamma(\gamma)},$$

- where R is the universal gas constant ($8,3144 \text{ J.mol}^{-1}.\text{K}^{-1}$),
- M is the molecular weight of the gas (kg/mole),
- T_o is the gas temperature (K),
- γ is the isentropic exponent (C_p/C_v),
- $\Gamma(\gamma)$ is the Vandenkerkhove's function: $\Gamma(\gamma) = \sqrt{\gamma} \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma+1}{2(\gamma-1)}}$

C^* characterizes obviously the combustion products and is totally independent of the expansion conditions and on the external pressure; nozzle chocking plays the role of an insulator between the combustion chamber and the external surrounding.

The third parameter, again defined for a chemical rocket and a chocked nozzle, is the thrust coefficient:

$$C_F = \frac{F}{p_o A_c} \text{ (non-dimensional)}$$

C_F depends on the expansion rate and the external pressure. The simplified description of the nozzle flow gives: $C_F = C_F^{(A)} + \Delta C_F$:

- where $C_F^{(A)}$ is the thrust coefficient for nozzle adaptation,
- ΔC_F is a correction of inadaptation.

It is possible to demonstrate that C_F is limited when the expansion rate increases. A simple relationship between the different characteristic parameters can be written: $I_s = \frac{C^* \cdot C_F}{g_o}$

This relationship gives a clear interpretation of the respective contributions of combustion products and nozzle in the thrust production. Thermochemical computation gives C^* , C_F and I_s by taking into account the actual thermodynamics of combustion products. For the conventional composite propellant previously mentioned:

$$\begin{aligned} C^* &\approx 1550 \text{ m.s}^{-1} \\ C_F &\approx 1,68 \quad (\text{adapted nozzle}) \end{aligned}$$

1.4 A First Approach of Motor Operation

A first approach of motor operation can be established by using the elementary *global* conservation laws. The main necessary assumptions are the following:

- Steady-state operation,
- Very low Mach number ($M \ll 1$) of the combustion products in the combustion chamber (that means that ratio of nozzle entrance section area to throat area is very large),
- The combustion products are totally generated on the combustion surface.

In these conditions, only two conservation principles for mass and energy should be satisfied:

- $\dot{m}_b = \dot{m}_T$, where \dot{m}_b is the mass flow rate of combustion products entering the combustion chamber and \dot{m}_T is the mass flow rate exhausted by the nozzle.
- $\dot{m}_b h_p = \dot{m}_T h$, where h_p is the specific enthalpy of combustion products and h is the specific enthalpy of the flow entering the nozzle.

Obviously $h_p = h$ and the stagnation temperature at nozzle entrance is the temperature of combustion products. Therefore it remains: $\dot{m}_b = \dot{m}_T$, where $\dot{m}_b = \rho_p r_b A_b$ (ρ_p : propellant density, r_b : burning rate, A_b : combustion area), $\dot{m}_T = \frac{p_o A_c}{C^*}$, following the definition of C^* .

A very simple relationship exists, expressing the mass conservation, between the different parameters:

$$\rho_p r_b A_b = \frac{p_o A_c}{C^*}$$

It is more appropriate to write: $\frac{p_o}{\rho_p r_b C^*} = \frac{A_b}{A_c}$.

The right hand expression is related to the *geometric* parameters of the rocket motor:

$$K_G = \frac{A_b}{A_c} \quad (K \text{ for the german word } Klemmung).$$

The left hand expression depends on the propellant characteristics (ρ_p, r_b, C^*) and on the pressure (p_o). The (normal steady-state) burning rate is a function of pressure, *for a given propellant*:

$$K_p(p_o) = K_G$$

This equality between the parameters K_p , function of p_o , for a given propellant, and K_G , based on the rocket geometry, can be used in different ways:

- Determine A_c si A_b and p_o are chosen,
- Predict p_o si A_b, A_c are fixed...

Finally, the first step of the design of a solid rocket propellant looks very simple. Unfortunately, is not sufficient to perform a complete motor design with the required high level of prediction and reliability, as explained later. However, the early solid propellant rocket motors have been developed on this basis.

1.5 Evolution of Parameters According to Time

A solid rocket motor has a variable internal geometry related to the continuous transformation of the solid propellant to combustion products. Change of the combustion area in time will result in change of pressure and thrust. A steady state view of operation can be adopted by considering different geometries separated by an increment of displacement of the combustion surface. An elementary analysis can justify this approach. One considers an elementary surface by assuming that the transformation of the solid propellant to combustion products occurs on the surface area and not in volume (fig. 3).

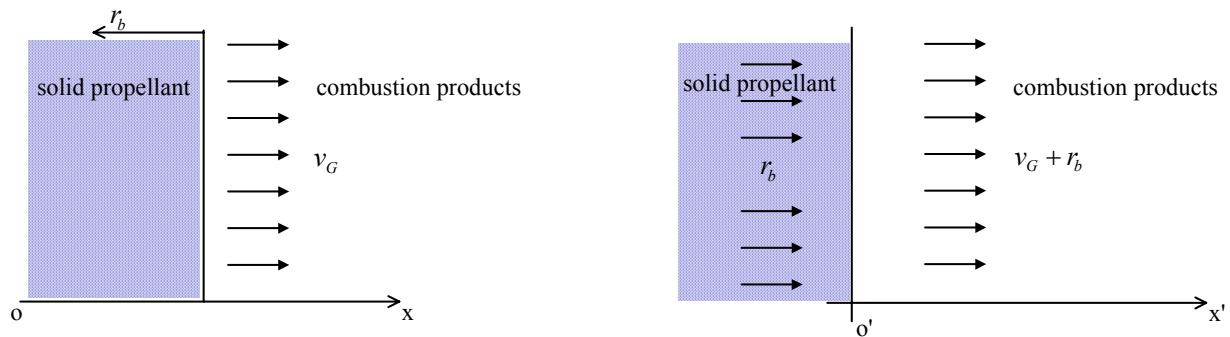


Figure 3: Injection Conditions on the Combustion Surface.

Mass conservation leads to:

$$\rho_G v_G \approx \rho_p r_b$$

If $\rho_p = 1,78 \text{ g.cm}^{-3}$, $r_b = 8 \text{ mm.s}^{-1}$, $p = 70 \text{ atm}$,

$T_o = 3494 \text{ K}$, $M = 28,46 \text{ g.mole}^{-1}$, one finds: $V_G \approx 2 \text{ m.s}^{-1} \gg r_b$

$$M_G \approx 1 \times 10^{-3}$$

Two consequences result from this elementary analysis:

- Velocity of the combustion products leaving the combustion surface is at least two order of magnitude higher than the burning rate; a reasonable approximation is to assume that the products are entering the combustion chamber from a fixed surface. This approximation justifies all the experiments performed by cold gas simulation (injection of a cold gas through a porous wall).
- Description of the flow in the entire motor implies Mach number from very low values (near the combustion surface) to very high values (nozzle exit) and then use of adapted numerical methods.

Conservation of momentum has also another consequence: the combustion products are emitted perpendicularly towards the combustion surface.

2.0 THE MAIN ISSUES OF SOLID PROPELLANT ROCKET MOTOR

Development of a solid rocket motor requires considerable technological investments in different domains. These investments are substantial given the severe conditions inside the motor: high pressure (around 5 MPa or 50 bar in the larger boosters), high temperature (around 3500 K for a composite metallized propellant), two phase flow (metallized propellant). The critical technologies concern the propellant itself, the propellant grain design, the thermal insulation, the nozzle design and the case. Besides, another well known difficulty is related to the near impossibility to perform detailed measurement inside the combustion chamber, due to the severe conditions prevailing there: the only usual measurement is the static pressure measurement through a pressure tube, generally at the motor front end. This difficulty can be considered as one of the main obstacles to the perfect validation of the prediction tools.

As this Short Course is devoted to aerodynamics, attention will now be focused on the particular aspects of design having an influence on aerodynamics and operation.

2.1 Burning Rate

Propellant burning rate plays a central role in motor operation. That is why perfect control of burning rate is absolutely mandatory. Fundamentals and recent developments regarding burning rate will be developed in another lecture. The discussion will here be limited to a classification of the different combustion regimes encountered during the whole motor operation.

2.1.1 Steady State Regimes

- Normal (regular) regime: burning rate is depending on only two parameters: pressure and initial temperature: $r_b = r_b(p_o; T_i)$. It is known that burning rate can change with the propellant temperature. Since only a very thin layer of the propellant is usually affected by the chemical transformation during the combustion process, a good assumption is that the initial grain temperature (considered as uniform) is an external parameter, without variation during operation. Sensitivity to pressure is related to the chemical reactions in gaseous phase (flames) above the combustion surface.

Some parameters are defined to characterize locally the propellant sensitivity to pressure and temperature:

$$n = \left(\frac{\partial \ln r_b}{\partial \ln p_o} \right)_{T_i} = n(p_o; T_i)$$

$$\sigma_p = \left(\frac{\partial \ln r_b}{\partial T_i} \right)_{p_o} = \sigma_p(p_o; T_i), (K^{-1})$$

It should be mentioned that the pressure exponent n should be lower than 1 for a practical propellant; if not, the steady-state operation pressure is not stable; $n \approx 0.3$ for the more conventional composite propellants.

- Erosive burning regime: if the flow above the combustion surface is fast, it is possible to observe an increase in the burning rate for a given pressure (and, less frequently, a decrease in the burning rate for some double base propellants). A general accepted explanation of such an increase is linked to the enhancement of the convective thermal flux to the combustion surface due to the interaction between flow turbulence and flames. In this condition, burning rate is no longer pure feature of the propellant but a property mixing propellant and flow. Erosive burning can be responsible of a pressure overshoot at ignition. A simplified prediction can be obtained by characterizing the flow by the unit mass flow rate in the perforation (internal burning tube); erosive burning is then a threshold phenomenon.

2.1.2 Unsteady Regimes

When pressure or more generally the conditions above the combustion surface are changing very rapidly, direct dependence of burning rate on instantaneous parameters is no longer right; time is also involved. Several situations bring into play unsteady burning rate, essentially the transient phases (ignition and operation end) and the instability. The common approach is to make the distinction between linear burning rate and non-linear burning rate.

Linear burning rate corresponds to small perturbations of all the parameters around a steady state and it is typically the case for the low level instability. Response to the propellant burning rate to pressure (pressure coupling) has been proved (measured and explained); the response can be considered like a transfer function between pressure and burning rate:

$$R_{MP} = \frac{\tilde{r}_b / \bar{r}_b}{\tilde{p}_o / \bar{p}_o}$$

where, for instance, \bar{p} is the steady state pressure and \tilde{p} the amplitude of oscillation.

R_{MP} is a complex parameter, function of the oscillation frequency. A temperature response R_{TP} can be derived from R_P . At relatively low frequency, the pressure response is only linked to the thermal transfer in the solid propellant. R_P plays an essential role in the combustion instabilities.

A response of the propellant burning to flow (velocity coupling) has been suspected but never really proved and measured.

Non-linear burning rate occurs either during ignition or during high level instabilities. If some analytical or computational tools are able to predict non linear burning rate, their validation by experiments is still very poor.

The figure 4 summarizes the occurrence of the different burning rate regimes.

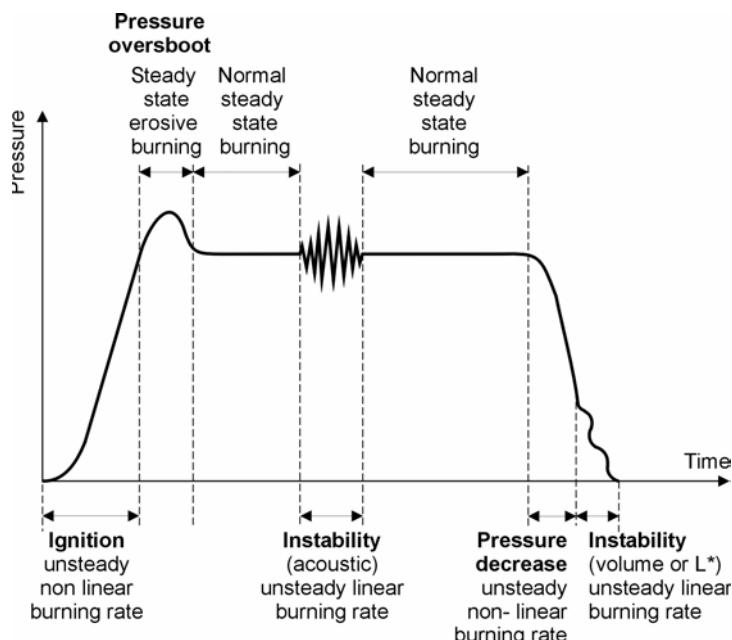


Figure 4: The Different Operation Regimes.

2.2 Grain Design

To achieve a better motor performance, an obvious objective is to integrate the maximum of propellant in a given case volume. The volumetric loading fraction can exceed 0.92 for a launcher upper stage. The thrust program should also be adapted to the motor mission: for the first stage of a space launcher, the thrust should be high at take-off then be reduced during the transonic flight and then can increase again; in the case of two large boosters attached to a central core (Titan 4, Ariane 5 for instance), attention should also be given to the synchronization of both thrust decreases before total extinguishment of the boosters. So a compromise should be found between different requirements. Art of the designer is to find the grain geometry leading to the better global performance, by satisfying a number of constraints at the same time.

Increasing the volumetric loading fraction leads to decrease the central bore and the area ratio at nozzle entrance, then with the direct consequence to increase the Mach number flow in the combustion chamber. This increase can produce erosive burning at the end of perforation and more generally generates differences of static pressure between the motor front end and the motor aft end; as the mechanical properties of a propellant are limited, particular care should be devoted to mechanical grain design to avoid any unexpected crack or dangerous grain deformation. Another consequence, more directly connected to aerodynamics, is that the simplified approach (§ 2.4) is not yet valid or accurate enough. Therefore refinement of flow description inside the combustion chamber is required.

3.0 IMPORTANCE AND NECESSITY OF SOPHISTICATED AERODYNAMIC PREDICTION

There are two main domains which need the development of the knowledge in internal aerodynamics: the steady-state operation and the transient unsteady regimes.

3.1 Steady-State Operation

Modern SRM design is characterized by a tridimensional geometry, generally associating axisymmetric and star-shaped patterns (for instance, finocyl grain). Grain grooves (French asymmetrical machined grains) or intersegments (segmented grains) generate noticeable pressure variation which cannot be predicted by too crude computation. Another trend is to use composite (filament winding) case, more deformable than a metallic one; the case deformation and the grain deformation should be taken into account in this situation and it is particularly important for the first phase of operation following ignition. So a natural trend is to couple the internal aerodynamics and the case/grain deformation, by taking into account the visco-elastic feature of the grain. Finally it appears that numerical computation can only provide an accurate prediction of the pressure field inside the combustion chamber, especially in the first part of the operation.

Currently, but with the exception of the CSAR program, prediction of steady operation is made by a set of computations at different fixed geometries, which are computed by another code. The principle is the following (fig. 5):

- Aerodynamics at t : aerodynamic computation for a known geometry giving the pressure field on the combustion surface and then the field of burning rate on the combustion surface.
- Geometry at $t + \delta t$: geometrical computation of the new geometry by using the field of burning rate on the combustion surface at t .
- Aerodynamics at $t + \delta t$: new aerodynamic computation...

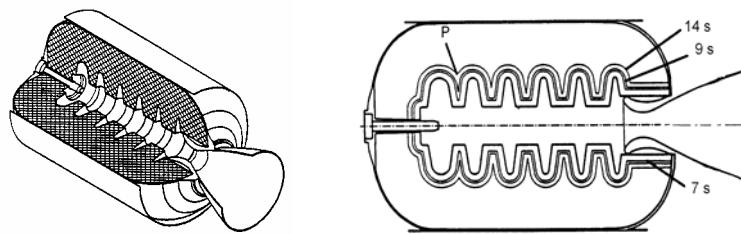


Figure 5: Grain Geometry Evolution.

A totally coupled computation of aerodynamics and geometry evolution requires use of variable meshes in the combustion chamber and in the grain and is very expensive in terms of computation time. In the classical approach previously mentioned, where accuracy is, *a priori*, of first order, δt can be chosen to maintain the computation time in a reasonable range.

It should be also mentioned that the modern codes do not distinguish the flow zones at low velocity and the flow zones at high velocity. The whole flow from the combustion surface to the nozzle exit is described and computed with the same compressible equations and the same numerical methods. This way of computing avoids difficult and unsolved issues of matching between domains (combustion chamber, nozzle).

3.2 Unsteady Regimes

The main issues are connected to the ignition phase and to the prediction of the stability (or of the instability level).

If certain characteristic times can be predicted by simple analytical calculations, using an unsteady variant of the global conservation laws, this type of approach is totally insufficient for a total mastery of the ignition phase. In fact, transient ignition phase should be described by taking into account a lot of phenomena: igniter operation, thermal convective and radiative transfer from the igniter combustion products to the inert grain combustion surface, propellant surface temperature increasing and propellant ignition, unsteady propellant combustion, flame propagation. In detail and for realistic igniter, grain and propellants, the complete problem is rather difficult to solve because of the propagation of various waves inside the motor, the difficulty to predict accurately the thermal transfer for igniter two phase combustion products and in the recirculation zones, and of the unsteady combustion. However some modern validated codes (unsteady two phase flow), are able to solve the problem.

More difficult is the prediction of stability or level instability. Instability is a natural phenomenon observed for a long time on all the engines, from the pressure recording: instead of being stable (with a little random component due to the noise), pressure exhibits an oscillation well organized around a frequency, or possibly around several frequencies to form a non-harmonic unsteady signal. That is the appearance of an instability. This phenomenon, called combustion instability or operation instability, is rather frequent and a lot of solid rocket motors are flying with a low level of instability during certain phases of their operation; but an instability can become very dangerous if its level increases: in this case the mean burning rate tends to be erratic, the pressure uncontrolled and the thermal insulation consumption drastically augmented.

Different classes of instability have been determined fifty years ago. The simplest one to understand is the volume instability or L^* instability; it is clearly a combustion instability, i.e. that the propellant pressure coupling is needed to drive the instability. The L^* instability occurs generally at low pressure and low frequency and is not dangerous. The most critical and scientifically interesting instabilities are acoustic:

that means that the unsteady pressure distribution inside the combustion has a certain relationship with the unsteady pressure distribution corresponding to a pure acoustic mode in the cavity which is defined by the combustion surface and a closure in the nozzle entrance plan. This type of instability is known as a combustion instability because propellant pressure response is conditioning stability in the classical analytical approach (acoustic balance). It is not the case in reality for the first longitudinal modes in the larger booster, the propellant pressure response seeming not to be the primary driving phenomenon; the term of operation instability is then, maybe, more appropriate. Stability should be assessed at each operation time, that is to say for different grain geometries (fig. 5).

Prediction of steady-state operation and unsteady regimes clearly requires a detailed description of the flow inside the combustion chamber. Evolution of the computation method (and of the computer capacities) has allowed a substantial progress during the last thirty years, even if this evolution does not yet solve all the problems.

4.0 APPROACH OF INTERNAL AERODYNAMICS IN SOLID ROCKET PROPULSION BY COMPUTATION

Approach of steady state operation by using the global conservation laws is under the dependency of different assumptions and finds rapidly its limits, either because of the difficulty to find explicit analytical solutions (case of ignition for instance) or because of the physics which needs a more local and refined flow description. Concerning this second point, an academic example can be mentioned, the one concerning the nozzleless motor (fig. 6). For a sufficient length to perforation diameter ratio and a rather high propellant burning rate, flow inside the perforation is naturally choked; the explanation is the formation of a multiple fluid nozzles in the flow due to the lateral injection.

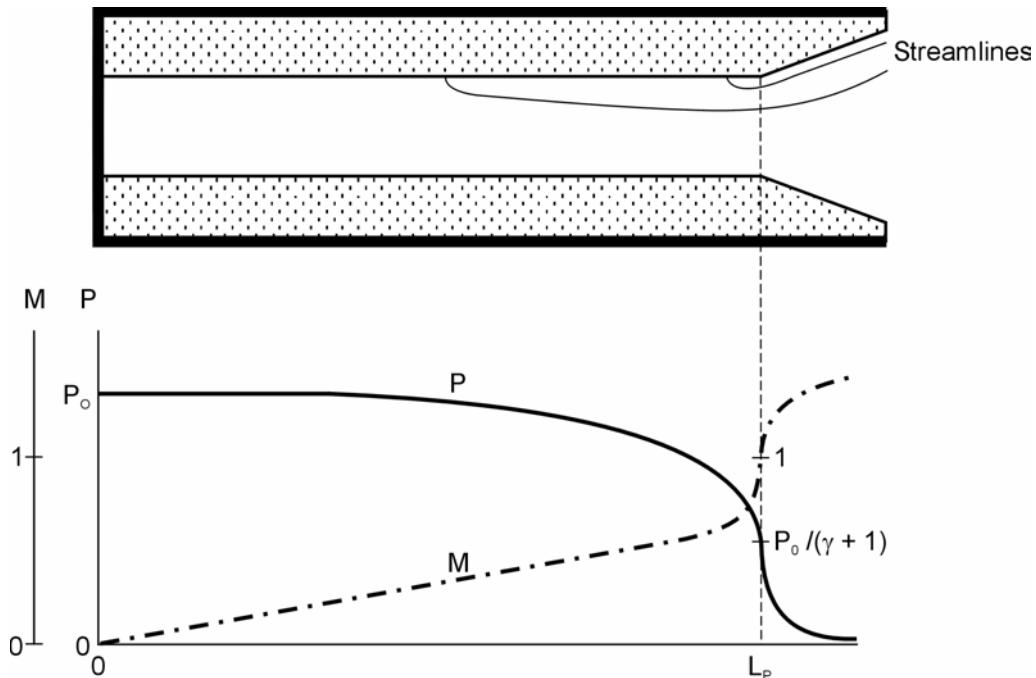


Figure 6: The Nozzleless SRM.

By using some simplifications (constant burning rate and constant diameter along the combustion surface, ideal gas), it is possible to determine the pressure at the front end (here p_o) and the distribution of

pressure and Mach number (1D description) in the perforation. It has been demonstrated that p_o can be calculated by the relation:

$$\frac{p_o}{\rho_p r_b(p_o) C^*} = 2 \left(\frac{2}{\gamma + 1} \right)^{\frac{1}{\gamma-1}} \frac{2L}{R}$$

or, under another form:

$$K_p(p_o) = 2 \left(\frac{2}{\gamma + 1} \right)^{\frac{1}{\gamma-1}} K_G,$$

where K_G is the ratio of the combustion area to the passage area A_p . The relation obtained is very similar to those previously given for a conventional motor with a large nozzle entrance ratio, except for the factor $2(2/(\gamma+1))^{\frac{1}{\gamma-1}}$. This difference is the direct consequence of the organization of the flow, the Mach number of which being 1 at the end of the cylindrical perforation. This example proves that aerodynamics can be essential for pressure prediction. There is total continuity between the conventional rocket operation and the nozzleless motor operation; it is possible to propose a generalization of the relation between $K_p(p_o)$ and K_G (always with some simplifications): $K_p(p_o) = k K_G$, where k can be calculated as the solution of a transcendental equation: $k(K_2, \gamma)$ where: $K_2 = A_p / A_c$

$$k \rightarrow 1 \text{ if } K_2 \rightarrow \infty$$

$$k = 2(2/(\gamma+1))^{1/(\gamma-1)} \text{ if } K_2 = 1$$

Real computation of the steady flow began at the beginning of the 70's for a one dimensional flow description, it was extended to a 2D flow description in the 80's, then to a 3D flow description in the 90's. The first computation codes have described the combustion products as an ideal gas (one-phase aerodynamics); extensions have been made but not always integrated in the code to take into account the actual thermodynamics (thermodynamic equilibrium or frozen composition) and to the two-phase flow aspects (combustion of aluminium droplets, alumina droplets). Use of the most advanced codes is still facing three main difficulties:

- Long computation time.
- Lack of accuracy of certain input data, especially for the condensed phases.
- Uncertainties on some phenomena: flow turbulence, interactions between droplets for instance.

Concerning the transient phenomena (ignition phase), some derivatives have been made from the previous codes without too many difficulties for the flow phenomena. The main issues have been rather connected to the heat transfer and the propagation of the flame on the combustion surface. However, some validations have been made by using a special small scale model with a dedicated instrumentation and the results are rather promising. Besides, the physical time to simulate is rather small and computation time affordable. Coupling between aerodynamics and grain/case deformation should be improved.

Prediction of the level of instability requires in essence a non-linear approach. Extension of the acoustic balance and coupling of modes, by the averaging method, constitutes a first possibility and presents the advantage of a clear physical understanding. In parallel, a numerical approach can be used and certain improvements have been made for the past ten years; some of them will be presented in lectures by F. Vuillot, G. Casalis and J.F. Guery. Without claiming to be the ideal solution, computation allows a

certain level of prediction in certain situations. The most current computations are two-dimensional, when that is allowed by the grain geometry, even if there is no intrinsic difficulty for 3D computations, but only some problems of computation time.

The main difficulties to face for instability level prediction are the same as those mentioned for the steady-state computation: lack of accuracy of certain input data (condensed phases...), uncertainties regarding the description of fundamental phenomena (flow turbulence, interaction between droplets...) but physics is here more complicated since unsteady. In fact, the situation can be summarized as follows:

- Rather good level of prediction at low scale for a non metallized propellant and sometimes simplified geometries, when the unsteady phenomena driving phenomena are well identified.
- Relative uncertainties at large scale, for a metallized propellant and complicated geometries; large sensitivity to input data.

5.0 PHYSICAL OBSTACLES

If computation is more and more used by industry, some physical obstacles remain to be overcome, in order to achieve a better prediction on actual motors and for all the regimes. Unsteady flow description can be considered today as the general needed approach, the steady state operation being viewed as the limit of an unsteady operation. The numerical methods should be adapted to correctly deal with the propagation of the different waves (acoustic, entropic, vorticity). In some specific situations, it has been proved that vortex shedding plays a central role in combustion instability but in other situations, it appears that other phenomena will prevail. Instability of tactical motors, especially for tangential modes, certainly gives more importance to pressure coupling and acoustic streaming.

The basic phenomena, whose description should be greatly improved, can be classified in three categories:

- Flow phenomena: turbulence in one-phase flow and two-phase flow. Turbulence generated by combustion is very poorly known. Two-phase phenomena are suspected to have an influence in some conditions (distributed aluminium combustion, wave damping by alumina droplets) but the data remains uncertain.
- Propellant combustion: unsteady combustion of the propellant remains a challenge, especially for a metallized propellant; for this type of propellant, it is not clear, for instance, if the usual definition of the linear response is totally valid.
- Fluid-“structure” coupling: structure is understood here as the assembly of all the solid parts of the motor: restrictors, grain, case, nozzle. Participation of restrictor vibration has been suspected for instability driving in some large segmented SRMs. But the problem is more general and invites to couple aerodynamics and solid mechanics.

The well known difficulty to improve the knowledge of this different phenomena is linked to the nearly impossibility to carry out detailed measurements inside an actual motor, even at a low scale and for a non-metallized propellant. The phenomena should be isolated and studied on skilled set-ups, for which the main issue remains to be their representativity.

6.0 PROSPECTS

This introduction has attempted to throw a bridge between the apparent simplicity of the rocket motors and the difficulty to perfectly master its operation; internal aerodynamics forms the pile of this bridge and cannot be bypassed.

Development of a large SRM needs a very large investment. For instance, development of the Ariane 5 has required around 2€ billion (including the propellant factory and the test bench in Kourou). Each ground test costs around 20€ million. Cost reduction is presently a leitmotiv for the civil and military industry. A general trend is therefore to develop reliable tools to reduce the number of tests during development and qualification. Tools are also needed to explain abnormal operation.

Since 1990, some important national programs have been initiated and have produced new results:

- In France, the ASSM program, with the sponsorship of CNES, under the scientific responsibility of ONERA and with the participation of several CNRS and universities laboratories (and of the VKI), for the Ariane 5 MPS P230 (SRM). Mainly focused on instability level prediction, this program allowed to develop numerical tools, to improve the flow description and... to measure the limits of the available knowledge. A new program is currently being prepared between France and Italy (SPADA program).
- In the USA, among the different known programs, the MURI program and the CSAR program. This last one, funded by the DoE is focusing on very high level computation, is more ambitious and more general than the programme ASSM and has started producing some initial encouraging results.

It is still a long way to solve all the issues regarding SRM aerodynamics. According to the author's opinion, a good balance should be maintained between computational improvement and improvement of the knowledge regarding the basic phenomena.

This short course provides a unique opportunity to bring together the specialists deeply involved in Europe and in the USA in SRM aerodynamics. All the important aspects of aerodynamics and SRM operation mentioned in this introduction are covered and will give to the attendees an updated state of the art.

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